RELEASE DATE
MAY 12 1958
CONFIDENTIAL
UNCLASSIFIED

Copy
RM E58A27b

NACA RM E58A27b

NACA

RESEARCH MEMORANDUM

INVESTIGATION OF AN UNDERSLUNG HALF-CONE INLET WITH COMPRESSION SURFACE MOUNTED OUTBOARD FROM FUSELAGE AT MACH NUMBERS OF 1.5, 1.8, AND 2.0

By Richard A. Yeager and Laurence W. Gertsma

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

CLASSIFICATION CHANGED
UNCLASSIFIED

MAY 1 . 1958

LANGLEY AERONAUTICAL LABORATORY
LIBRARY, NACA
LANGLEY FIELD, VIRGINIA

To

By authority of TPA #33 10-28-60
ERG

CLASSIFIED DOCUMENT

This material contains information affecting the National Defense of the United States within the meaning of the espionage laws, Title 18, U.S.C., Secs. 793 and 794, the transmission or revelation of which in any manner to an unauthorized person is prohibited by law.

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON
May 12, 1958

CONFIDENTIAL
UNCLASSIFIED

3 1176 01435 9047

[redacted] UNCLASSIFIED

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

INVESTIGATION OF AN UNDERSLUNG HALF-CONE INLET WITH COMPRESSION SURFACE MOUNTED OUTBOARD FROM FUSELAGE AT MACH NUMBERS OF 1.5, 1.8, AND 2.0

By Richard A. Yeager and Laurence W. Gertsma

4742

CZ-1

SUMMARY

An investigation was conducted to determine the performance of an underslung half-cone inlet mounted on a missile forebody model with the compression surface outboard from the fuselage. The inlet was designed for shock-on-lip operation at Mach number 2.0 with a 25° half-angle spike. The cowling was attached to the fuselage through the boundary-layer plow and served as part of the fuselage boundary-layer diverter system.

The performance of the half-cone inlet was compared with that of a scoop-type inlet (ref. 1) and a normal-wedge inlet (ref. 2) on a maximum-thrust-minus-drag basis. The increase in pressure recovery obtained with the half-cone inlet over that obtained with the reference inlets offset the slightly higher drags observed over the Mach number range for the half-cone so that the performance of this configuration was equal to that of the other inlets at Mach number 2.0 and was slightly superior at the lower Mach numbers. For a particular configuration, a peak pressure recovery of 0.879 was obtained at Mach number 2.0, zero angle of attack, and 4-percent throat bleed; the subcritical stability was 16 percent. Use of a fuselage-mounted boundary-layer splitter plate ahead of the inlet did not improve the stability. Subcritical distortion values were below 10 percent for all configurations.

INTRODUCTION

In comparison with conventional side inlets that have compression surfaces contiguous with the fuselage, inlets having compression surfaces outboard from the fuselage tend to provide less cowl drag surface and conceivably lower diffuser-exit air distortion, since these inlets avoid turning the flow first away from the fuselage and then back into the engine compressor. References 1 to 3 report studies of inlets with outboard compression surfaces where the compression was essentially two-dimensional.

[redacted] UNCLASSIFIED

The inlets were studied as bottom inlets on a model of a missile forebody. As an extension of these studies, a half-cone inlet providing three-dimensional supersonic compression has been investigated on the same forebody model. This inlet was designed for shock-on-lip operation at Mach number 2.0 with a fixed spike of 25° half-angle. The cowling was attached to the fuselage through the boundary-layer plow and thus acted as a part of the fuselage boundary-layer diverter system in an attempt to reduce the sum of the cowl pressure drag and the drag associated with boundary-layer removal.

4742

The investigation included a study of the effects of throat bleed, several inlet approaches, a fuselage boundary-layer splitter plate ahead of the inlet, and a cone floor plate over a range of angles of attack from -5° to 15° at free-stream Mach numbers of 1.5, 1.8, and 2.0 in the Lewis 8- by 6-foot supersonic wind tunnel. In addition, an over-all thrust-minus-drag comparison between the present three-dimensional half-cone inlet and the inlets reported in references 1 and 2 was made.

SYMBOLS

A	area, sq ft
A_{in}	inlet capture area, 0.1506 sq ft
A_{ref}	reference area (body maximum cross-sectional area), 0.915 sq ft
C_D	drag coefficient based on A_{ref}
D	full-scale forebody drag, lb
D_b	full-scale bypass drag, lb
F_n	net thrust, lb
$F_{n,i}$	ideal net thrust (100-percent pressure recovery), lb
$\dfrac{F_n - D - D_b}{F_{n,i}}$	net-thrust-minus-drag ratio
h	minimum distance between cowl lip and fuselage
M	Mach number
m_3/m_0	ratio of mass flow at model station 97.6 to mass flow at free-stream conditions through inlet capture area A_{in}

4742

CZ-1 back

P total pressure, lb/sq ft

δ boundary-layer thickness

Subscripts:

av average

max maximum

min minimum

0 free stream

3 compressor-face station, model station 97.6

APPARATUS AND PROCEDURE

Model Details

The missile forebody model was sting-mounted in the Lewis 8- by 6-foot supersonic wind tunnel and is shown schematically in figure 1. Two different flat approaches to the inlet were investigated, one yielding an h/δ of 2.0 and the other an h/δ of 1.4 at Mach number 2.0. The h/δ of 2.0 configuration was used to isolate the inlet from fuselage effects as much as possible in order to obtain the basic inlet performance and was not intended to be a practical configuration.

The inlet was designed for shock-on-lip operation at Mach number 2.0 with a fixed half-cone spike of 25° mounted outboard from the fuselage, as shown schematically in figure 1. Photographs of the inlet appear in figure 2. The cowling was attached to the fuselage through the boundary-layer plow and thus acted as a part of the fuselage boundary-layer diverter system. A flush slot was located in the half-cone surface just inside the cowl to remove the compression-surface boundary layer. This boundary-layer air was bled through a chamber and spilled back into the free stream by means of a variable bypass door, the details of which are shown in figures 1 and 2(c). Figure 1 also shows duct cross sections from the cowl lip to the compressor hub-tip station.

In an attempt to reduce the effects of the interaction of the inlet normal shock with the fuselage boundary layer during subcritical operation and thus improve the subcritical stability range of the inlet system, a fuselage boundary-layer splitter plate (figs. 2(a) and (c)) was strut-mounted to the fuselage just upstream of the cowl lip for part of the investigation. Also, for part of the test, a cone floor plate (fig. 2(b)) was employed in an attempt to decrease the amount of supercritical spillage. The fairing of the inlet lines aft from the cowl lip into the fuselage (fig. 2(c)) was not necessarily optimum with respect to drag, since the inlet was adapted to an already existing forebody. Subsonic diffuser area variations are shown in figure 3.

Instrumentation and Data Reduction

Eight equally spaced total-pressure rakes were located at the compressor-face station. Each rake consisted of five area-weighted total-pressure tubes. 4742 Static-pressure orifices were located on the duct wall and centerbody at the ends of each rake. Pressure recovery and flow distortion were based on the average of the area-weighted total-pressure tubes. Mass flow was controlled by varying a plug at the duct exit. Just upstream of the duct exit, eight static-pressure orifices were located, four in the duct outer wall and four in the centerbody. Mass-flow calculations were made by using the average static pressure obtained from these orifices, with the assumption of a choked geometrical minimum area determined at the duct exit by plug position.

Axial and normal forces were measured by a combination of an internally mounted strain-gage balance system located forward in the model and a rear normal-force link located at the aft bulkhead. Forces measured by the balance system were the combined internal duct forces, external fuselage forces, and base forces. The drag presented is the streamwise component of the measured forces excluding the base force and the change in momentum of the internal flow from free stream to the duct exit.

The test was conducted over a range of angles of attack from -5° to 15° at free-stream Mach numbers of 1.5, 1.8, and 2.0.

RESULTS AND DISCUSSION

Effect of Throat Bleed on Inlet Performance

By varying the bypass door position, the amount of bleed through the flush slot in the half-cone surface was varied. In subsequent discussion the designation of the amount of bleed refers to the bleed mass flow at critical operation at Mach number 2.0 and is expressed in percent of the free-stream reference mass flow m_0. Increasing the amount of bleed had only a small effect on pressure recovery, as can be seen in figure 4. Four-percent throat bleed increased the peak pressure recovery from 0.885 (no-bleed case) to 0.895 at Mach number 2.0. Increasing the throat bleed further had no effect on pressure recovery, but gains in subcritical stability were obtained. There was little effect on distortion with bleed; no distortion values existed above 9 percent in the subcritical stable range. Upon close examination a very slight decrease in drag appears to exist by opening the bypass door; this is probably due to the more favorable body fairing resulting when the bypass door was open. The position of the bleed door that yielded 4-percent throat bleed at Mach number 2.0 and zero angle of attack was held constant throughout the remainder of the test.

Effect of Angle of Attack on Inlet Performance

4742

Presented in figure 5 are the effects of angle of attack on inlet performance. The peak pressure recovery increased with increasing angle of attack over the Mach number range investigated, while at negative angles of attack the recovery was less. Increasing angle of attack caused increased distortions at Mach number 2.0, while at Mach number 1.8 little effect was observed. At Mach number 1.5, lower distortion values were obtained with increased angle of attack. The drag decreased with increasing angle of attack up to 5° over the Mach number range. Above 5° the drag increased rapidly. It should be noted that the drag values presented in figure 5 are somewhat high, since they were obtained with the h/δ of 2.0 configuration, where the fuselage boundary-layer diverter system was handling an amount of air in excess of that required to yield good inlet performance; however, the effects of angle of attack were the same with both configurations.

Effect of Inlet Approaches

The effect of inlet approach on inlet performance is shown in figure 6. The two configurations investigated are designated by the h/δ of each determined at Mach number 2.0. There was a slight increase in peak pressure recovery when the h/δ of 2.0 configuration was employed instead of the h/δ of 1.4, but this increase was only 1 percent at Mach number 2.0 and somewhat less at the other Mach numbers. There was little effect on distortion over the Mach number range; thus it appears that the effects of interaction of the inlet normal shock with fuselage boundary layer for a more practical h/δ of 1.4 were small. The reduction in drag obtained by employing the h/δ of 1.4 configuration was directly associated with the smaller amount of air handled by the fuselage boundary-layer diverter system.

Effect of Fuselage Boundary-Layer Splitter Plate Ahead of Inlet and Effect of Cone Floor Plate

In an attempt to reduce the effects of interaction of the inlet normal shock with the fuselage boundary layer and to improve the subcritical stability range, a fuselage boundary-layer splitter plate was mounted ahead of the inlet. As shown in figure 7, no improvement in subcritical stability range was obtained by employing the splitter plate; and the distortion was slightly increased, especially at Mach number 1.5. No effect on drag was observed with the plate in position.

In an attempt to decrease the supercritical spillage, a cone floor plate was used for part of the test. The effects of the cone floor plate are shown in figure 8. Although data are shown only for Mach number 2.0, similar trends were observed at the lower Mach numbers. Small increases in peak pressure recovery were obtained with little effect on distortion. 2747
At zero angle of attack, the drag was slightly higher with the cone floor plate employed.

Thrust-Minus-Drag Analysis

In order to compare the scoop-type inlet of reference 1 and the normal-wedge inlet of reference 2 with the present half-cone inlet on the basis of a single performance parameter, a net-thrust ratio including a bypass drag $\frac{F_n - D - D_b}{F_{n,i}}$ was determined. These net-thrust computations were made by assuming that a typical turbojet engine was matched to a fixed-size inlet with a sonic bypass discharging air parallel to the free stream. The largest value of this parameter for each inlet at each Mach number at 5° angle of attack is plotted in figure 9. The half-cone inlet with no throat bleed yielded performance equal to the reference inlets at the higher Mach numbers and slightly better at the lower Mach numbers. This performance level was obtained because of the higher pressure recoveries obtained with the half-cone inlet offsetting the slightly higher drags observed over the Mach number range. These drags possibly could be reduced by a more favorable fairing of the inlet lines aft into the fuselage; the present fairing was not optimum from a drag consideration since the present inlet was adapted to an existing forebody. Because of further increases in pressure recovery, the half-cone inlet with 4-percent throat bleed yielded better performance than either the no-bleed configuration or the reference inlets over the Mach number range. It should be noted that the reference inlets had no throat-bleed arrangements.

SUMMARY OF RESULTS

An underslung half-cone inlet configuration with the compression surface outboard from the fuselage was investigated on a missile forebody model. The inlet was designed for shock-on-lip operation at Mach number 2.0 with a fixed spike of 25° half-angle. The cowling was attached to the fuselage through the boundary-layer plow and thus acted as a part of the fuselage boundary-layer diverter system. The results obtained were compared with a previously tested scoop-type inlet and a normal-wedge inlet on a maximum-thrust-minus-drag basis. The investigation was conducted over a range of angles of attack from -5° to 15° at free-stream Mach numbers of 1.5, 1.8, and 2.0. The following results were observed:

1. The increase in pressure recovery obtained with the half-cone inlet over that obtained with the scoop-type and normal-wedge inlets offset the slightly higher drags observed over the Mach number range for the half-cone so that, on a thrust-minus-drag basis, this inlet gave performance equal to the other inlets at a Mach number of 2.0 and slightly superior at the lower Mach numbers.

2. Peak pressure recovery of 0.879 was obtained for a particular configuration at Mach number 2.0, zero angle of attack, and 4-percent throat bleed. For these conditions, a critical drag coefficient of 0.155 was obtained.

3. At Mach number 2.0 and zero angle of attack, the subcritical stability was 16 percent. Use of a fuselage-mounted boundary-layer splitter plate ahead of the inlet did not improve the stability.

4. Subcritical distortion values were below 10 percent for all configurations.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, February 12, 1958

REFERENCES

1. Weinstein, Maynard I., Vargo, Donald J., and McKevitt, Frank: Investigation of an Underslung Scoop Inlet at Mach Numbers to 1.99. NACA RM E56L11, 1957.

2. Vargo, Donald J., and Weinstein, Maynard I.: Investigation of an Underslung Normal-Wedge Inlet at Free-Stream Mach Numbers from 1.50 to 1.99. NACA RM E56F27, 1957.

3. Valerino, Alfred S., and Zappa, Robert F.: Investigation of Mass-Flow and Pressure-Recovery Characteristics of Several Underslung Scoop-Type Inlets at Free-Stream Mach Numbers of 2.0, 1.8, 1.5, and 0.66. NACA RM E56K29, 1957.

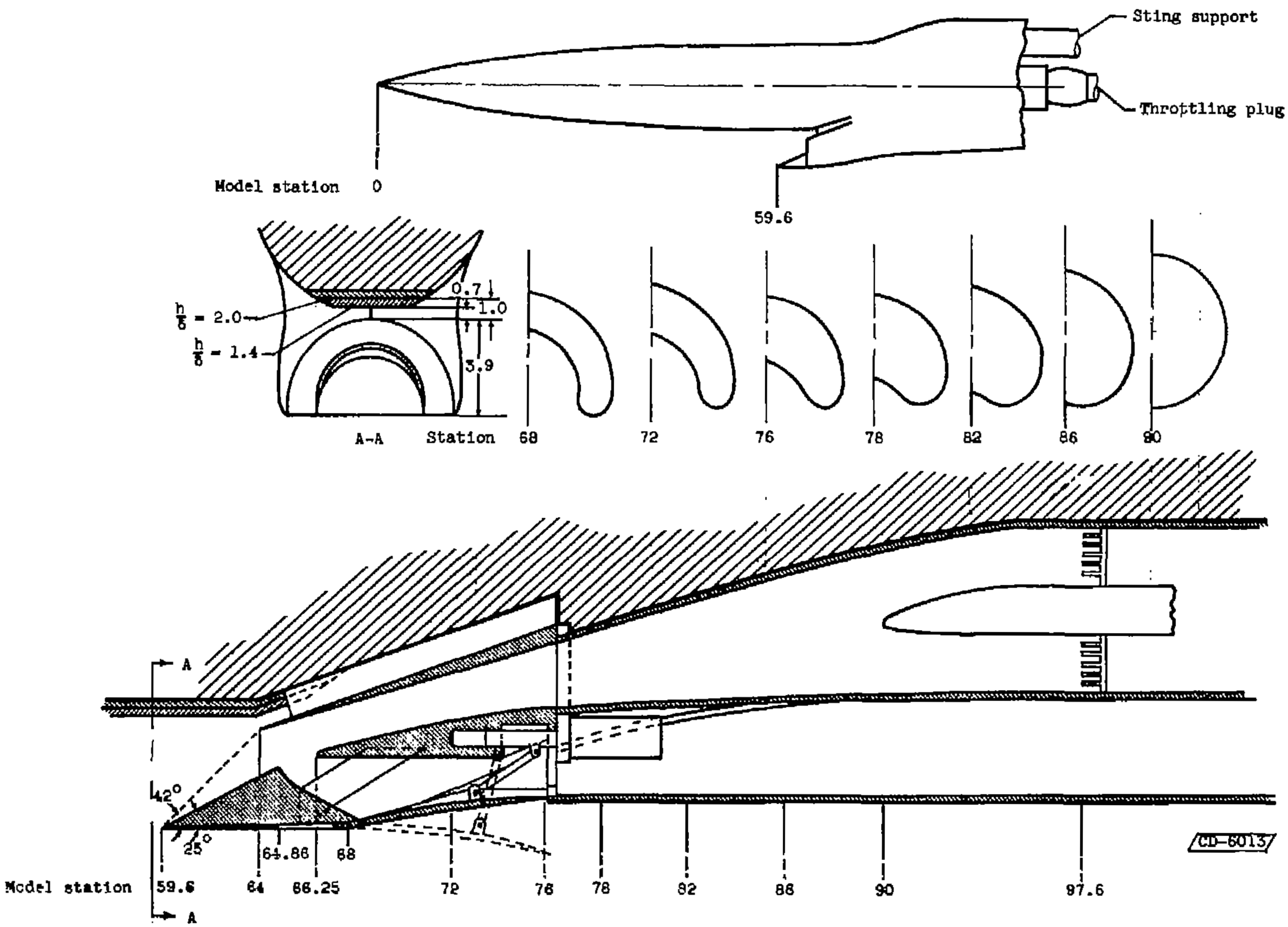
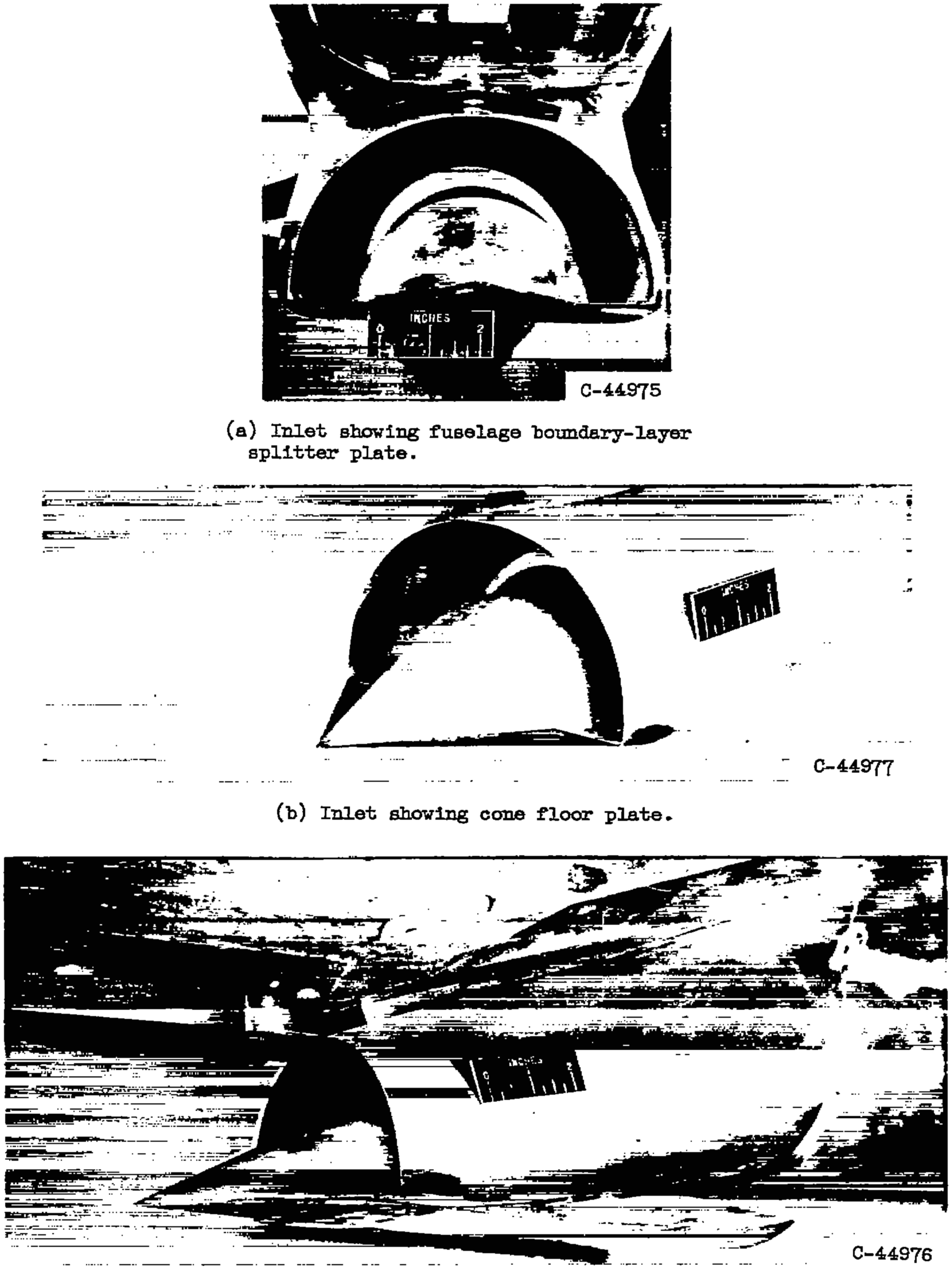


Figure 1. - Diagram of model. (All dimensions in inches.)

4742

CZ-2



C-44975

(a) Inlet showing fuselage boundary-layer splitter plate.

C-44977

(b) Inlet showing cone floor plate.

C-44976

(c) Inlet showing fuselage boundary-layer splitter plate and throat-slot bypass door open.

Figure 2. - Inlet model.

[redacted]

[redacted]

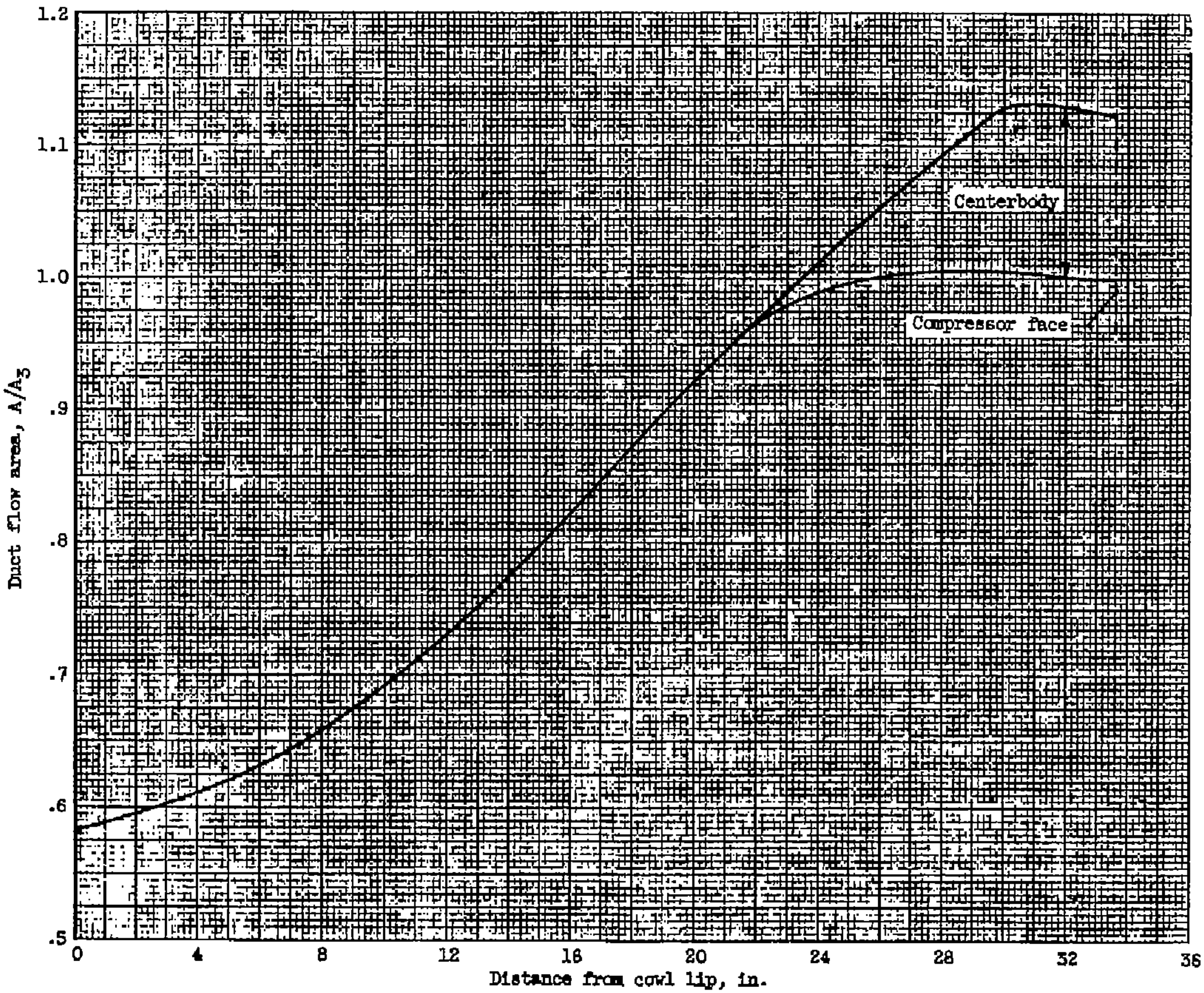


Figure 3. - Diffuser-area variation.

[redacted]

4742

CZ-2 back

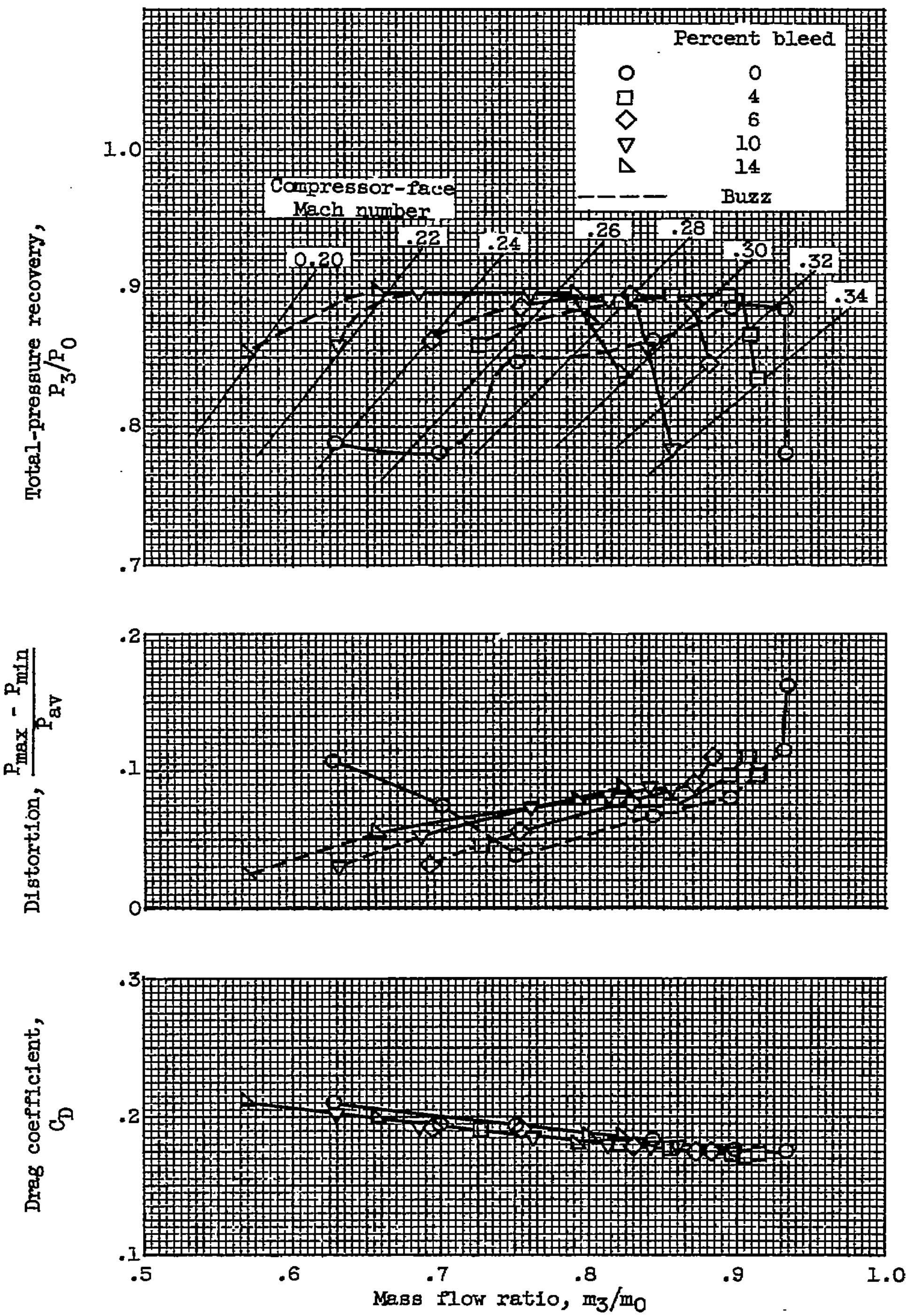


Figure 4. - Effect of throat bleed on inlet performance. Mach number, 2.0; angle of attack, 0°; h/δ, 2.0; with cone floor plate.

C[redacted]

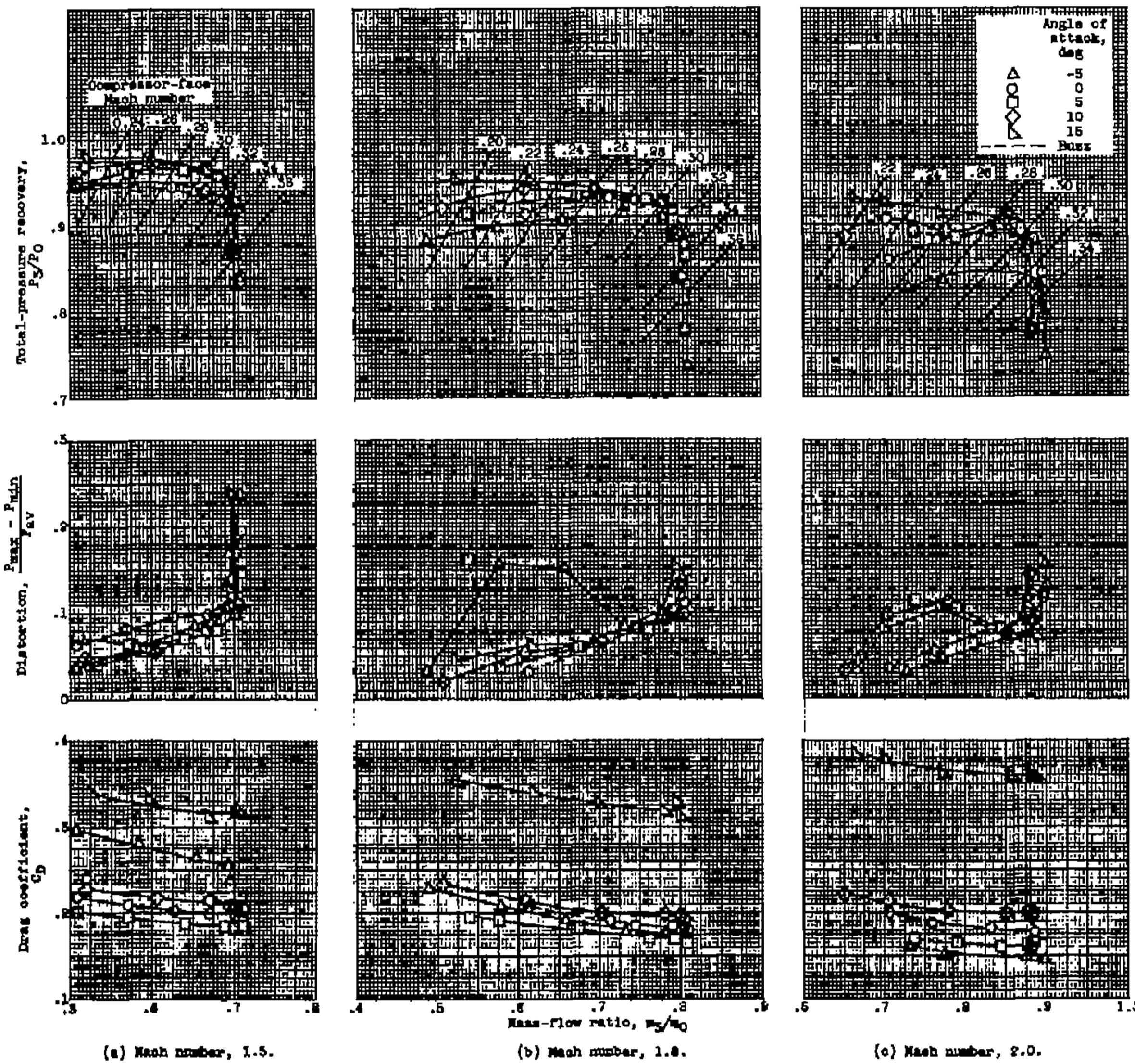
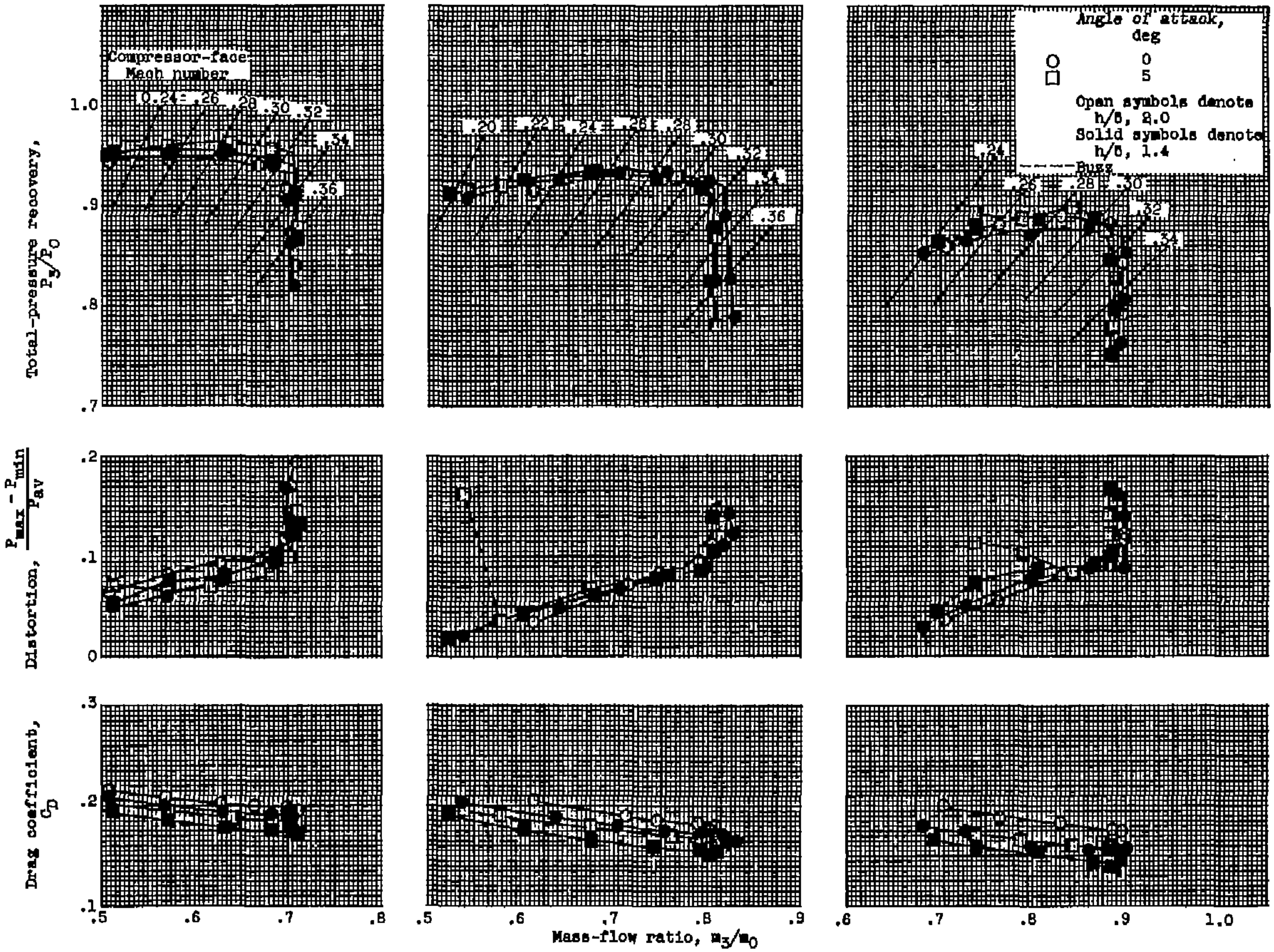


Figure 5. - Effect of angle of attack on inlet performance. Throat bleed, 4 percent; h/δ, 2.0.

[redacted]



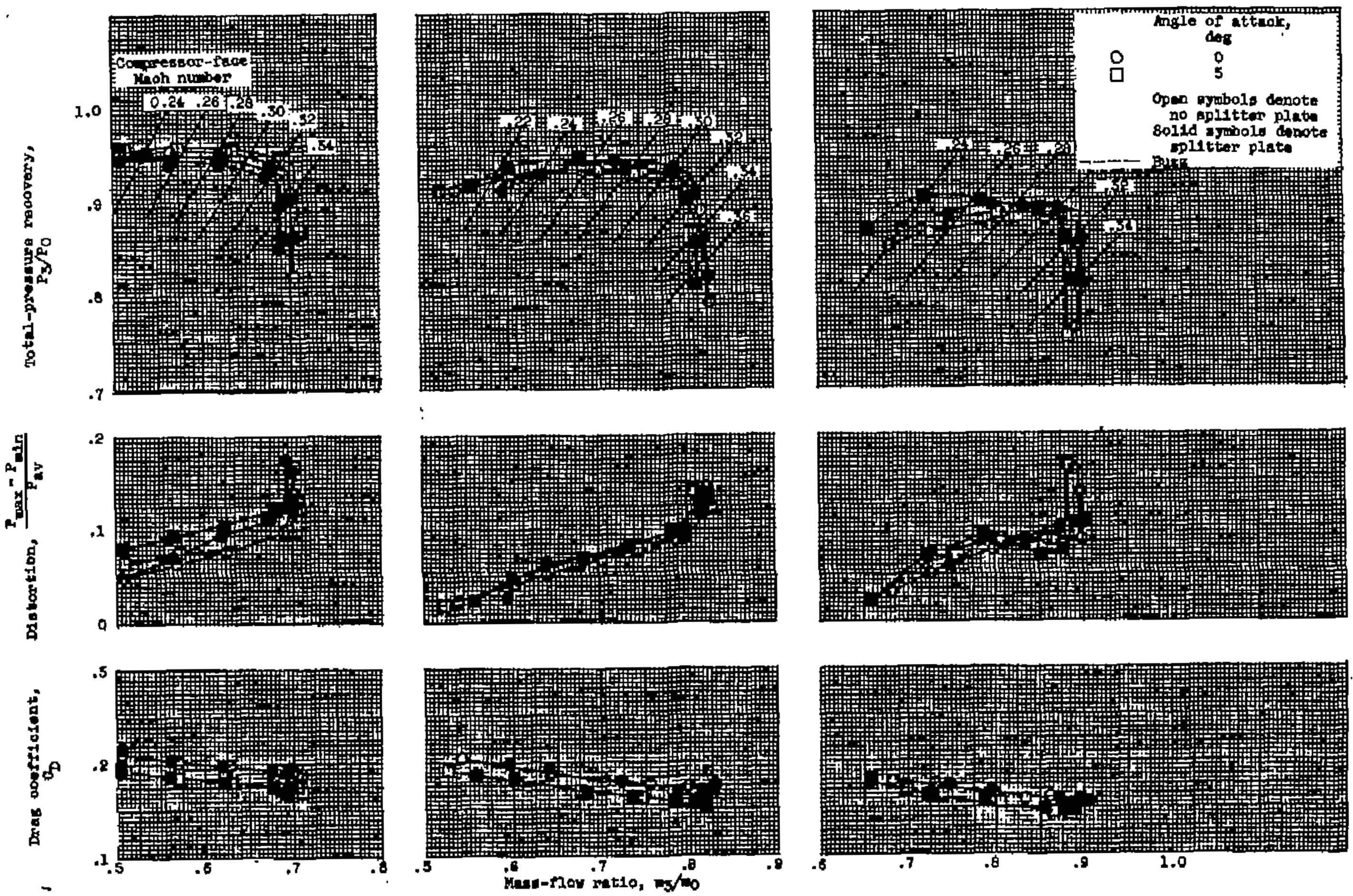
(a) Mach number, 1.5.

(b) Mach number, 1.8.

(c) Mach number, 2.0.

Figure 6. - Effect of inlet approach on performance. Throat bleed, 4 percent.

C[redacted]



(a) Mach number, 1.5.

(b) Mach number, 1.8.

(c) Mach number, 2.0.

Figure 7. - Effect on inlet performance of adding a fuselage splitter plate ahead of cowl. Throat bleed, 4 percent; h/δ, 1.4.

4742

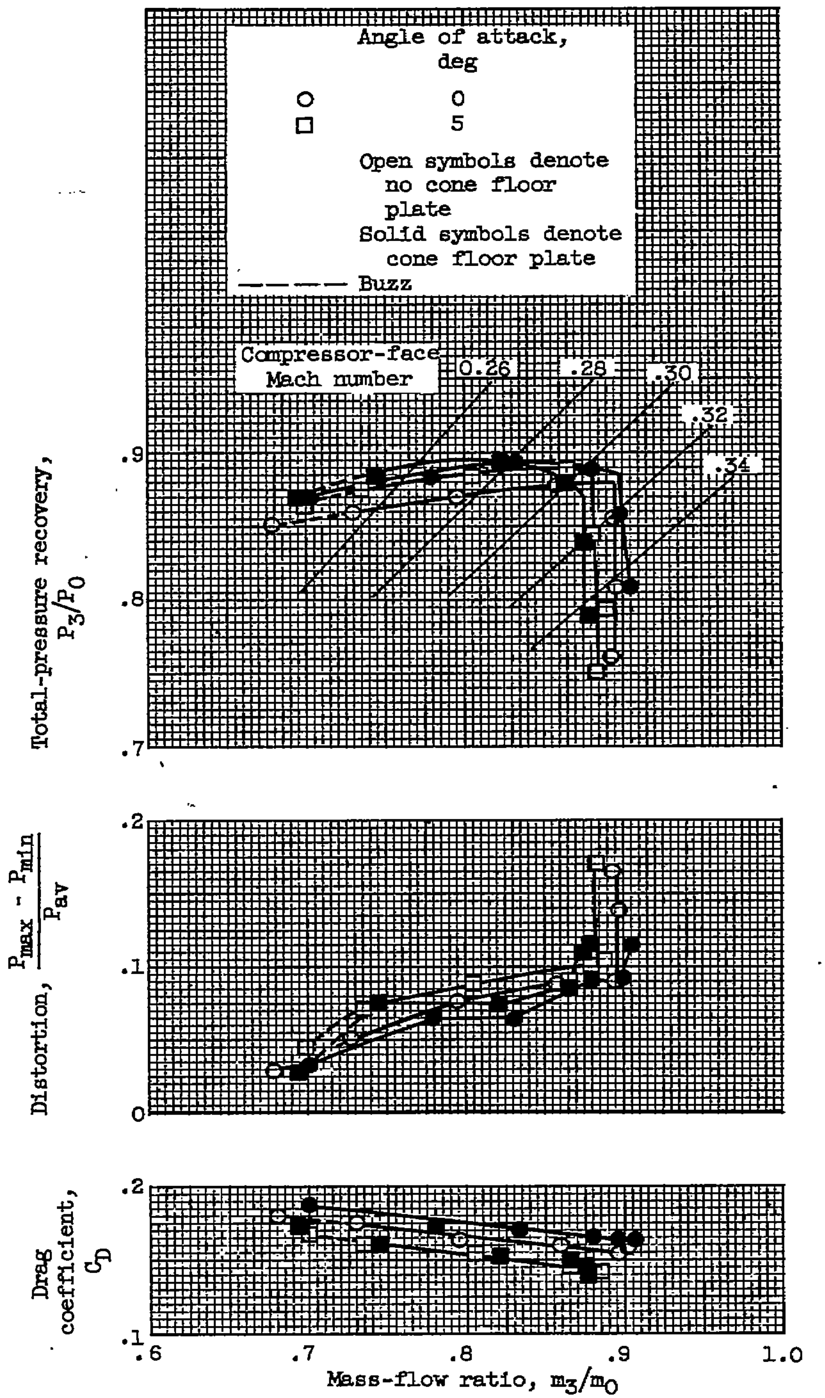


Figure 8. - Effect on inlet performance of adding a cone floor plate. Throat bleed, 4 percent; h/δ, 1.4; angle of attack, 0°; Mach number, 2.0.

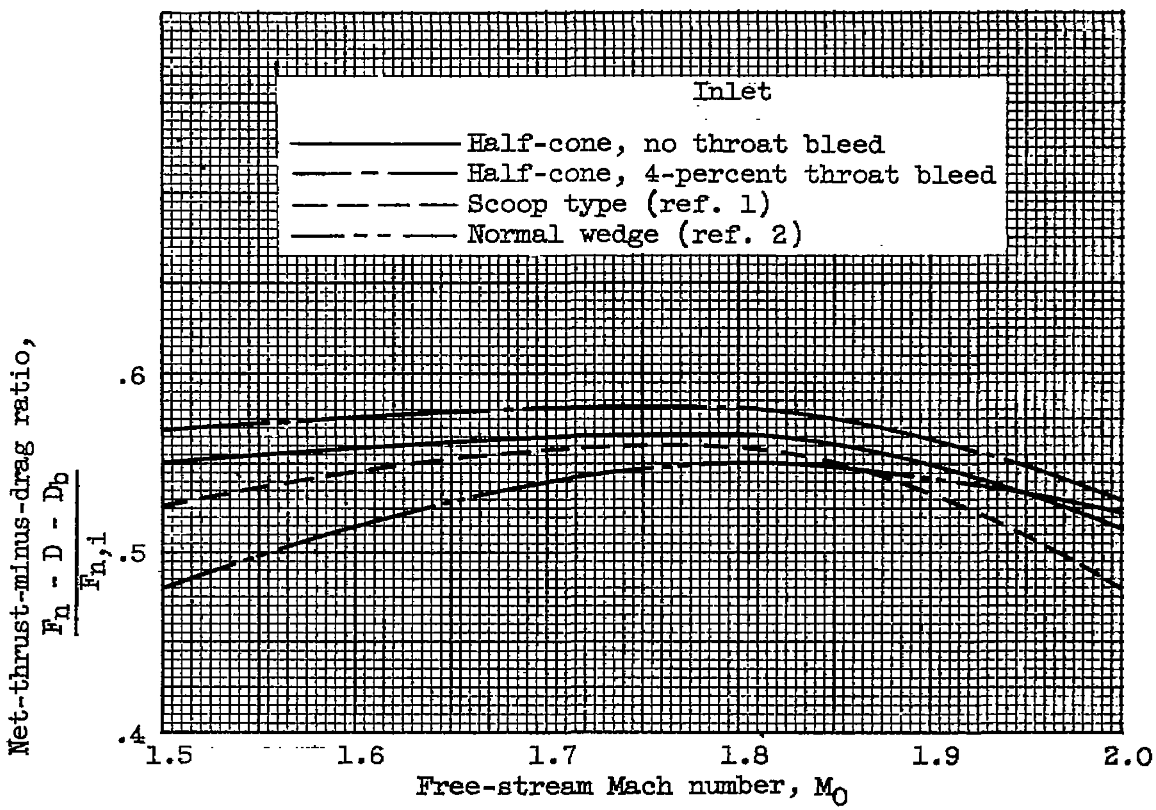
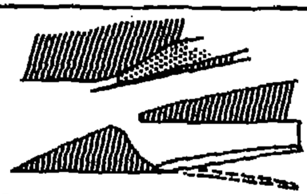


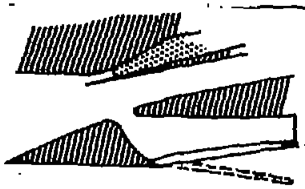
Figure 9. - Performance comparison of half-cone inlet with scoop-type and normal-wedge inlets. Angle of attack, 5°.

[redacted]

NOTES: (1) Reynolds number is based on the diameter of a circle with the same area as that of the capture area of the inlet.

(2) The symbol * denotes the occurrence of buzz.

Report and facility	Description: Configuration	Number of oblique shocks	Type of boundary-layer control	Test parameters: Free-stream Mach number	Reynolds number $\times 10^{-6}$	Angle of attack, deg	Angle of yaw, deg	Test data: Drag	Inlet-flow profile	Discharge-flow profile	Flow picture	Performance: Maximum total-pressure recovery	Mass-flow ratio	Remarks
CONFID. E58A27b wis 8- 6-ft personic nd nnel		1	Flush throat slot	2.0	2.16	-5,0,5,10, and 15	0	✓				0.895	0.7 - 0.9	Half-cone compression surface mounted outboard from missile forebody model fuselage. Thrust-minus-drag performance comparable to that of conventional design inlets.
				1.8	2.16	-5,0,5,10, and 15	0	✓				.930	0.5 - 0.81	
				1.5	2.16	-5,0,5,10, and 15	0	✓				.950	0.5 - 0.71	

Report and facility	Configuration	Number of oblique shocks	Type of boundary-layer control	Free-stream Mach number	Reynolds number $\times 10^{-6}$	Angle of attack, deg	Angle of yaw, deg	Drag	Inlet-flow profile	Discharge-flow profile	Flow picture	Maximum total-pressure recovery	Mass-flow ratio	Remarks
CONFID. M E58A27b ewis 8- y 6-ft upersonic ind unnel		1	Flush throat slot	2.0	2.16	-5,0,5,10, and 15	0	✓				0.895	0.7 - 0.9	Half-cone compression surface mounted outboard from missile forebody model fuselage. Thrust-minus-drag performance comparable to that of conventional design inlets.
				1.8	2.16	-5,0,5,10, and 15	0	✓				.950	0.5 - 0.81	
				1.5	2.16	-5,0,5,10, and 15	0	✓				.950	0.5 - 0.71	

Bibliography

These strips are provided for the convenience of the reader and can be removed from this report to compile a bibliography of NACA inlet reports. This page is being added only to inlet reports and is on a trial basis.

[redacted]